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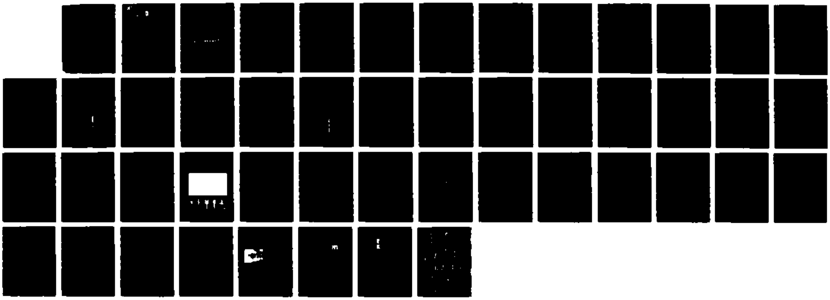

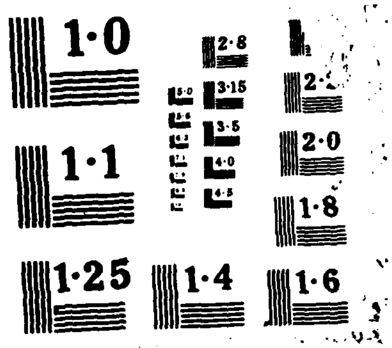
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This report describes an SBIR Phase I effort to determine the technical feasibility of measuring the performance of a pulsed electrothermal thruster on liquid hydrogen propellant. The design of a 5 kW thruster is presented, including the capacitive pulse forming network, arc discharge characteristics, nozzle performance and predicted specific impulse and thrust. A thermal model is discussed which treats and solves the problem of liquid hydrogen boiling in the injection nozzle. After an investigation of techniques for handling liquid hydrogen it is concluded that a Phase II test should concentrate on water and/or hydrazine, and liquid hydrogen testing should be postponed to Phase III.

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Strategic Defense
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Innovative Science and Technology
SBIR Phase I

Final Report

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AFOSR Contract No: F49620-87-C-0028DEF

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I. INTRODUCTION

This report provides an overview of a Phase I SBIR effort to determine the feasibility of performing liquid hydrogen performance tests on the pulsed electrothermal (PET) thruster in a laboratory environment. As part of this effort, the thruster was designed and its performance predicted.

It has been well established that electric propulsion has an important role to play for SDIO in space. The high specific impulse capability of electric thrusters has been demonstrated by electrothermal, electromagnetic and electrostatic devices possessing I_{sp}'s in the 500 to 5000 second range. In this section we derive the appropriate range for SDIO missions, and demonstrate that the pulsed electrothermal (PET) thruster proposed here has significant advantages for these missions.

What types of missions are best suited to electric thrusters, where they can perform a task which simply cannot be economically accomplished by chemical rockets? There are two types. The first type is a mission with a tight mass budget, e.g. a space-based radar which must be launched from the space shuttle and transferred to a higher parking orbit. Depending on the required orbit transfer, it may not be possible to accomplish the mission chemically and stay within the shuttle payload limit. In this case a high I_{sp} electric system can be used to achieve orbit transfer and minimize propellant consumption [1]. Because the thrust time increases as I_{sp}^2, the mass savings are ultimately dictated by the maximum allowable thrust time. For example, a 5000 sec electrostatic thruster may maximize on-orbit mass, but might require 400 days to complete the transfer. Therefore a 1000 second system might be preferable to reduce thrust time.

The second type of missions well suited to electric propulsion are those with high characteristic velocity V, conducted by reusable spacecraft. Examples of this type of mission are:

1. Transfer a satellite to GEO and return, or return a second satellite for maintenance, using an OTV.

2. Maneuver a battle station in a constantly changing orbit to avoid "space mines" and ASAT systems.

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[1] Burton, R.L., Goldstein, S.A., Hilko, B.K., Tidman, D.A., and Winsor, N.K., "Proposed System Design for a 20 kW Pulsed Electrothermal Thruster," AIAA Paper 84-1387, AIAA/SAE/ASME 20th Joint Propulsion Conference, Cincinnati, OH, June 11-13, 1984.

3. Transfer new battle stations or surveillance satellites into a constellation to build up the constellation or to replace spacecraft unexpectedly lost by malfunctions.

These missions are characterized by a requirement to move the maximum mass in the shortest possible time, in a reusable mode. An approach to finding the optimum I_{sp} for these missions is to define a payload rate [2]

$$\text{Payload rate} = PR = \frac{m_{pay}/m_o}{t_t}$$

where t_t is the thrust time. By maximizing PR, maximum use is gotten out of the spacecraft.

The payload rate PR is given approximately by the nondimensional expression:

$$\alpha(\Delta V)^2\ PR = 2\eta e^{\frac{-2\Delta V}{g\ I_{sp}}}\left(1 - \frac{\Delta V}{g\ I_{sp}}\right)\left(\frac{\Delta V}{g\ I_{sp}}\right)^2$$

where α[kg/w] is the power supply specific mass and η is the electric propulsion system efficiency. This expression has a maximum value for $\Delta V/gI_{sp}$ between 0 and 1. The optimum I_{sp} to maximize PR is given by the condition:

$$e^{-\Delta V/g\ I_{sp}}\left(1 - \frac{\Delta V}{g\ I_{sp}}\right) = m_p/m_o$$

where m_p/m_o is the power supply mass fraction. This equation is plotted in Fig. 1.

Figure 1 shows the following:

1. An I_{sp} of 500-3000 seconds is required for a wide range of missions from ΔV = 4-12 km/sec and power supply fractions m_p/m_o of 0 to 50%.

2. Payload mass fraction is a unique function of power supply mass fraction m_p/m_o. Payload fraction is approximately 35% for mp/mo up to 30%., even at a high mission ΔV of 12 km/sec and above.

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[2]Free, B., "Use of SP-100 Nuclear Power Supply for Electric Propulsion," Scionics, Inc., Tech, Rpt. STR No. 686, June 6, 1986.

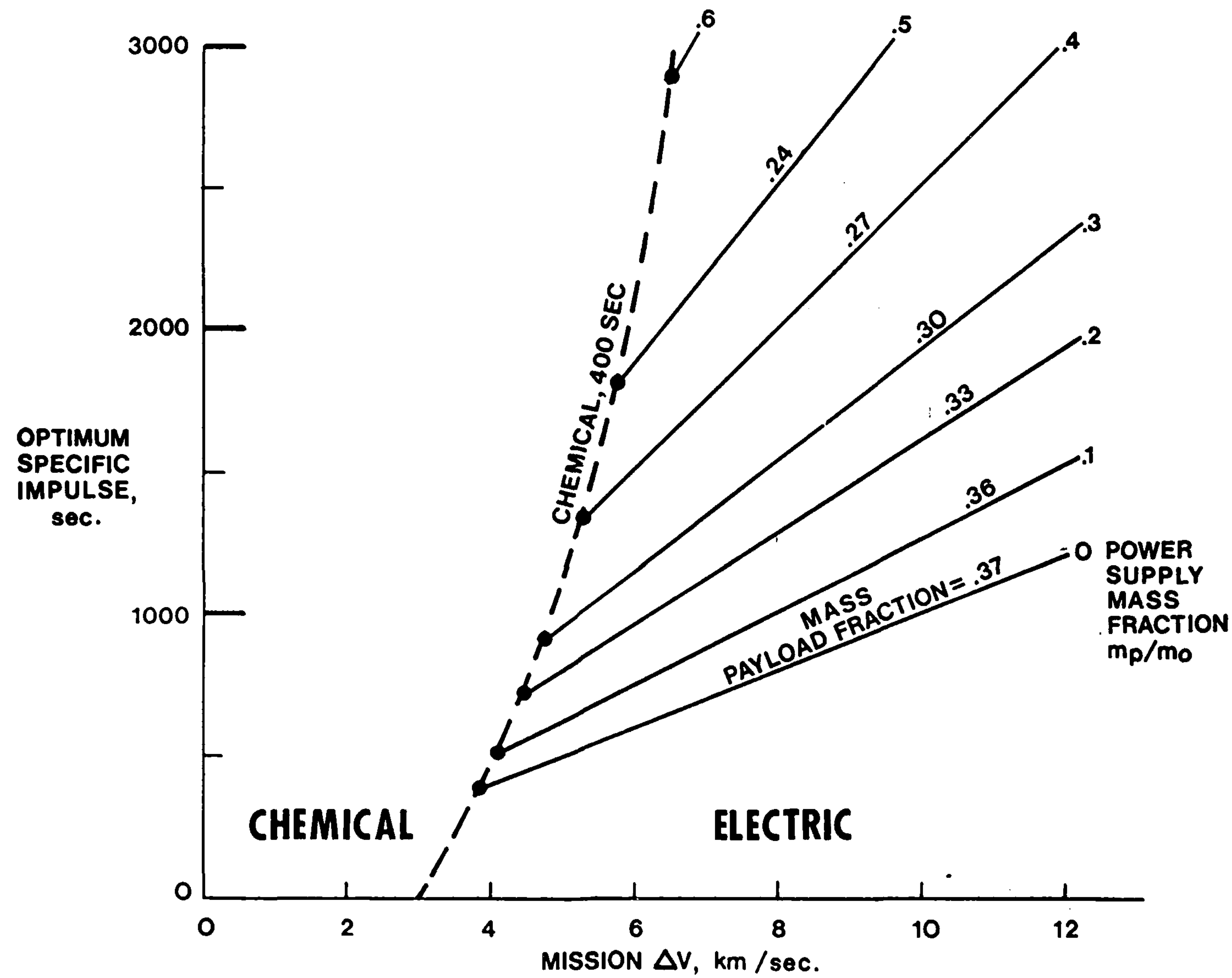

Figure 1. Optimum electric thruster I_{sp} to maximize payload rate as a function of ΔV and power supply mass fraction. The payload mass fraction is also shown. The dashed line shows where a 400 second chemical system provides a higher payload mass fraction than an electric system.

3. At mission ΔV's below 4 km/sec, the payload mass fraction for a 400 sec chemical system is higher than electric, so chemical is preferred over electric to the left of the dashed curve.

4. An I_{sp} of above 3000 seconds is only useful for very high ΔV's, well above 12 km/sec.

We now discuss the availability of electric thrusters in the 500-3000 second range, which is the desirable range for high payload rate as shown in Fig. 1. Electric thrusters have been under development since the early 1960's, and the performance status is shown in Fig. 2, which shows system efficiency versus I_{sp} for various thruster types. Beginning with the ion thrusters, the efficiency is above 80% at 5000 seconds, but falls off rapidly below 3000 seconds. This is a reflection of the constant energy cost that must be paid for each ion, which becomes a large (and unrecoverable) fraction of the energy below 3000 seconds.

Electrothermal thrusters (e.g. arcjet, Fig. 2) also produce ions, but unlike electrostatic ion thrusters can in principle recover their ion production costs in a nozzle, provided that nozzle recombination rates are sufficiently high to approach equilibrium. The recombination rates for the arcjet are not sufficiently high, as is reflected in the arcjet curve (Fig. 2), which shows a sharp dropoff in efficiency as the I_{sp} approaches 1000 seconds. This drop is directly related to the low recombination rate at the arcjet operating pressure of 1-2 atmospheres.

In order to recover the ionization energy and some of the dissociation energy in an electrothermal thruster, the recombination rate is raised by raising the plasma pressure; an effective remedy because the recombination rate varies as the square of the density [3]. Suppose that the pressure is raised by a factor of approximately 10^3, and the plasma energy is therefore raised by the same amount. To avoid raising the power level, which would seriously ablate the discharge chamber surfaces, the thruster is operated in a pulsed mode with a duty cycle of approximately 10^{-3}, which keeps the power throughput constant. This has the effect of bringing a kilowatt-level thruster into the megawatt range of peak power. The effect on efficiency is shown for H_2O and H_2 in Fig. 2, which indicates that efficiencies in the 80% range can be achieved with the H_2 at 2500 seconds, and 60% can be achieved with H_2O at 1500 seconds.

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[3] Book, D.L., "Plasma Physics," Chap. 18 in <u>Physics Vade Mecum</u>," H.L. Anderson, ed., Am. Inst. of Phys., New York, 1981, p. 271.

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ELECTRIC PROPULSION PERFORMANCE

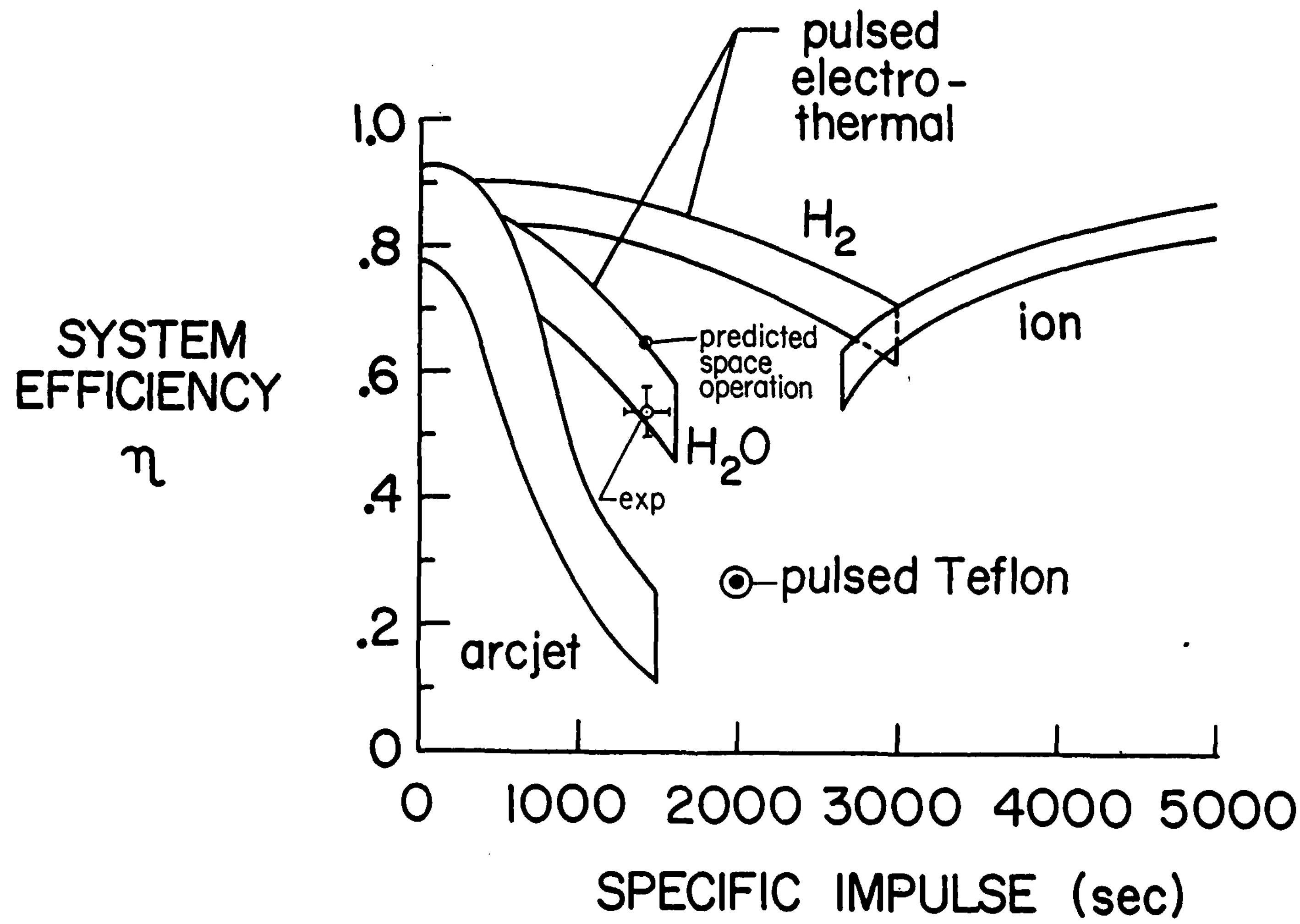

Figure 2

Pulsed Electrothermal (PET) Thruster Description

GT-Devices has been developing [4-7] the propulsion concept called pulsed electrothermal or PET. The system is shown in Fig. 3 in its simplest form. A power supply continuously charges an electric energy store, while liquid propellant is fed into a thrust chamber. The energy store then repetitively discharges electrical energy into the thrust chamber to create thrust impulses.

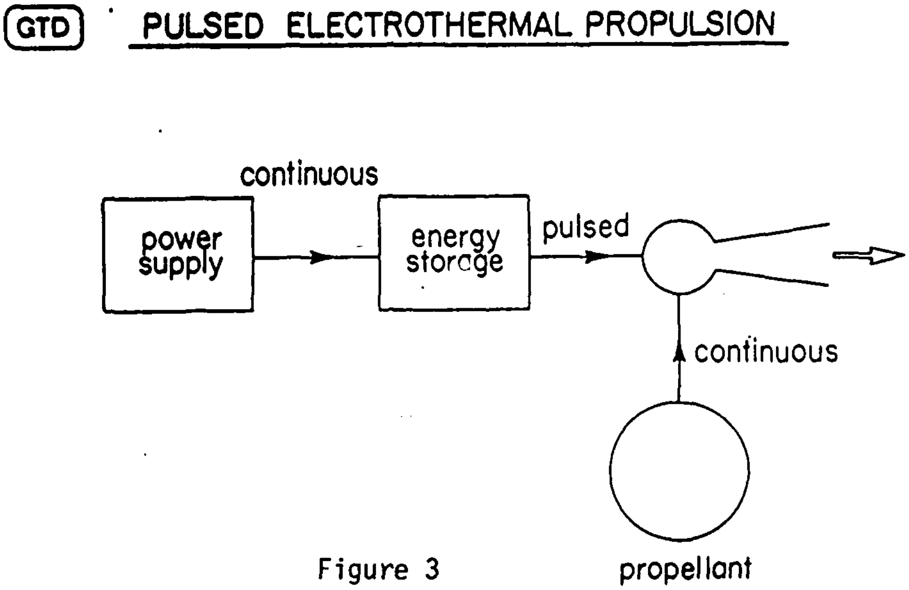

Figure 3

At first glance this system seems needlessly complicated, since the energy could be fed continuously to the thrust chamber to produce continuous thrust. This is in fact done in the

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[4] Burton, R.L., Goldstein, S.A., Hilko, B.K., Tidman, D.A., and Winsor, N.K., "Investigation of a Pulsed Electrothermal Thruster," Final Report No. NASA CR-168266, NASA-Lewis Research Center, October 1, 1983.

[5] Burton, R.L., Goldstein, S.A., Hilko, B.K., Tidman, D.A., and Winsor, N.K., "Investigation of a Pulsed Electrothermal Thruster System," Final Report on NASA CR-174768, NASA-Lewis Research Center, October 31, 1984.

[6] Burton, R.L., Fleischer, D., Goldstein, S.A., Tidman, D.A., and Winsor, N.K., "Investigation of a Repetitive Pulsed Electrothermal Thruster, Final Report No. NASA CR-179464, NASA-Lewis Research Center, August 21, 1986.

[7] Burton, R.L., P.I., "Performance of a Hydrogen Pulsed Electrothermal Thruster," SBIR Phase I, AFOSR Contract No. F49620-87-C-0028, 1/30/87 - 7/30/87.

hydrogen arcjet, but the efficiency decreases rapidly because of frozen flow losses above I_{sp} = 1000 seconds as shown in Fig. 2. The pulsed system not only increases the efficiency, but permits operation out to 3000 seconds (Fig. 2). This startling improvement in performance is serendipitously caused by a fundamental change in the physics of the nozzle expansion process, coupled with lower temperature operation, both of which substantially reduce the energy lost to ionization and dissociation in the exhaust. Our experimental measurements [6] have verified this performance improvement for water, and we predict an even greater improvement for hydrogen as shown in Fig. 2.

A PET propulsion system is shown schematically in Fig. 4. A full-wave rectifier charges a capacitive PFN which is directly connected to the electrodes. Simultaneously, a small quantity of propellant is injected into the chamber in liquid form. When the voltage is sufficiently high, breakdown occurs through the propellant vapor, and energy is efficiently transferred to the propellant, which has 99% of the circuit resistance. The propellant is heated to about 500 atm and 14,000°K, and expands through the nozzle, exhausting at a low temperature. The ionization energy of the exhaust is negligible, and the dissociation energy is low, so that the efficiency of the thruster is therefore quite high.

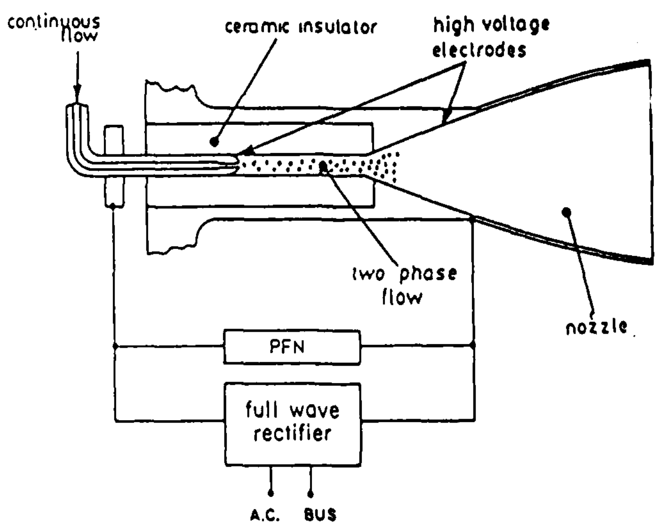

Figure 4

A pulsed system can be operated at an arbitrarily high power level, and hence pressure level, as long as the pulse length is sufficiently short that the electrodes and insulators do not

erode. It is straightforward to scale PET propulsion over a wide thrust range by varying the energy per pulse and the pulse rate.

The pulsed electrothermal thruster curves in Fig. 2 show two data points, which are estimated performance levels from preliminary tests on H_2O, conducted with considerable excess propellant flow at a (high) background pressure of 10-20 torr. It is the main purpose of the proposed effort to provide proper pulsed electrothermal thrust performance data, by conducting thrust stand tests on two or more propellants with a high propellant utilization efficiency.

It is important that realistic operating conditions exist during the performance tests. Simultaneously, it is required that:

1. The background pressure is sufficiently low to provide accurate exhaust velocity measurements and an ambient pressure considerably less than nozzle exit pressure.

2. Plasma conditions are such that the thruster does not erode or ablate.

3. The propellant is suitable for the 500-3000 second range, and the utilization efficiency is 50% or greater.

4. The performance measurements are accomplished with an accuracy of 2% or better.

Achievement of the above conditions and experimental results will establish that the PET thruster is capable of the performance required for high I_{sp}, minimum trip time missions for SDIO.

Phase I Effort

GT-Devices has recently completed the Phase I SBIR effort on performance of a liquid hydrogen PET thruster[7]. The two primary objectives were the design of a liquid hydrogen thruster and a determination of the feasibility of operating with LH_2 in the laboratory. A description of these two tasks and the conclusions reached form the body of this report.

II. DESIGN OF A LIQUID HYDROGEN PET THRUSTER

The design of the liquid hydrogen pulsed electrothermal thruster consists of three major elements:

A. Calculation of discharge parameters and pulse forming network.
B. Calculation of exhaust conditions, including frozen flow.
C. Thermal modeling of thruster, including boiling of liquid hydrogen prior to injection.

The thruster design is shown in Figure 5. Some of the hardware, such as a mount system and radiation shielding, is not included in this design.

Discharge Parameters, Pulse Forming Network, and Ideal Performance

Calculation of discharge parameters and pulse forming network follows a zero-D model used successfully at GT-Devices [4-6]. For this model, the discharge power is equated to the enthalpy flux of evaporated propellant, where the temperature is calculated self-consistently using a form of the Spitzer equation [8], modified for electron-neutral collisions [9]. The arc plasma is the load for a matched PFN, and the calculation is assumed to be quasi-steady.

The entire calculation is conveniently performed on LOTUS, and the present design is shown below:

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[8] L. Spitzer, Jr., "Physics of Fully Ionized Gases," 2nd Ed., Interscience Publishers, 1962. pp 136-143.

[9] D. A. Tidman, et.al., "A Rail Gun Plasma Armature Model," Tech. Note GTD86-3, March 1986.

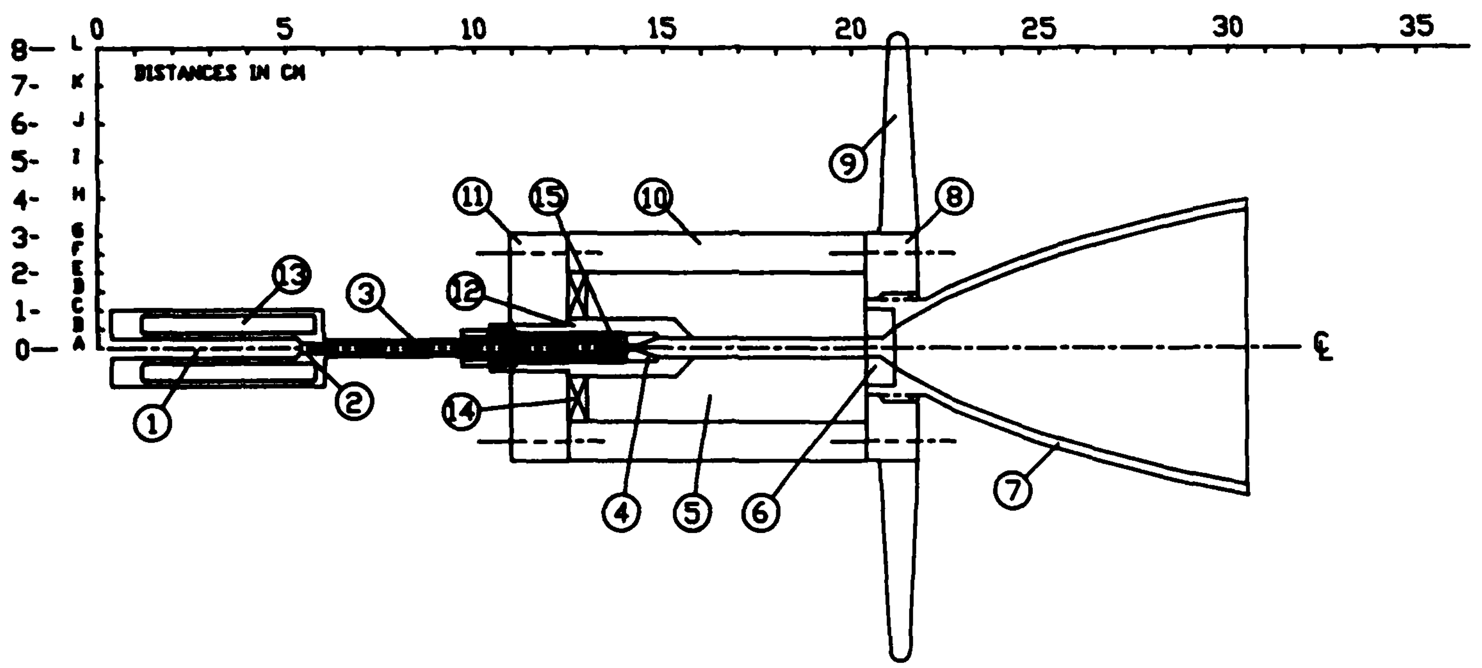

Figure 5. 5 kW LIQUID HYDROGEN PET THRUSTER

Electrothermal Model for PET   LH2PET   Revision date Sep 22,1987

This model calculates conditions in a bore being driven by electrothermal plasma sources, or "lightbulbs," and coupled to a supersonic nozzle.

I. Calculation of the conditions in the lightbulbs.

Lightbulb plasma parameters are calculated from the formulary. Equations in the formulary are presented as a function of current. In the model below the current is solved self-consistently from:

Vo = I(Zpfn + Rext + R(I))

where R(I) is derived from the formulary by eliminating Tev. One then gets an approximate quadratic equation of the form:

Y*Y + B*Y - Vo/(Zpfn + Rext) = 0

where Y = I^.5 and B is a constant. The value of B is given by:

B =((.0194)*.94*l*Sp^(8/11)*S^(3/11))/(Zpfn+Rext)/d^(13/11)/G

where Sp is the Spitzer term Zlnlambda(1+2nueo/nuei), S is the area enhancement factor, d is the capillary diameter and G is the volume factor. The variable l is the capillary length and for two lightbulbs in series is the sum of the two lengths. The .94 factor is 4(kA) raised to the -(1/22) power.

		ENTER?
Capillary length, cm.	3.50	E
Capillary diameter, cm, at 1 kilobar	0.45	E
Number of capillaries, 1 or 2.	1.00	E
Spitzer factor, Zlnlambda(1+2nueo/nuei)	8.00	E
Area enhancement factor, S	2.00	E
Volume factor, G	1.00	E
Impedance of pulse forming network, Zpfn, milliohms	900	E
Resistance of external circuit, Rext, milliohms	3	E
Capacitance of pfn, microfarads	8.00	E
Pulse length, microseconds, 2CZpfn	14.40	
Breakdown field, volts/cm	700	E
Charging voltage, Vo, kV	2.45	
PFN energy stored, J	24.01	
System average power, kW	5.00	E
Pulse rate, pps	208	
Constant B	1.00	
Current in capillary, kA	1.50	
Capillary resistance, milliohms, total	735	
PFN transfer efficiency, 4RZ/(R+Z)^2	0.99	
Second half cycle voltage, kV	0.25	

Capillary power, megawatts, total	1.64	
Capillary temperature, eV and deg K, T1	1.14	13276

In order to calculate pressure, information must be entered for the speed of sound and gamma, available from graphs or tables.

Material in capillary.	hydrogen	E
Molecular weight, STP	2.0	E
Speed of sound at capillary temperature, km/sec	11.90	E
Specific heat ratio, gamma	1.28	E
Throat area contraction ratio	0.45	E
Mach number (assumed sonic)	0.89	
Pressure, average, atm	100	
Pressure, stagnation, atm	128	
Pressure, sonic, atm	71	
Characteristic venting time, microseconds	5.54	
Pulse length, microseconds	14.40	
Enthalpy, kJ/gm, from tables	615.00	E
Ideal thrust impulse bit, mN-sec	1.35	
Ideal exhaust velocity, km/sec	35.07	
Ideal specific impulse, sec	3579	
Ideal thrust efficiency, eta	1.00	
Ideal thrust to power ratio T/P, N/kW	0.057	

Calculate the insulator wall heating, assuming blackbody radiation at the capillary temperature during the pulse, using the relation

dT = alpha*Tev^4*tp^.5

where alpha is a property of the insulator material. Typical values of alpha at high temps are 1.0 for SiC, 1.2 for BN and 1.4 for Si3N4. Melt temperatures in deg. C are 2500 for SiC, 3000 for BN and 1900 for Si3N4.

Also calculate the temperature rise due to convection for phi = 1.5. The MTF data give dT/alpha values of 3530 for SiC, 2530 for BN and 3003 for Si3N4. Handbook values are 2500, 2500 and 1360.

Insulator alpha	1.40	E
Insulator temperature rise, delta T, deg K.	911	
Delta T convective at the throat, deg K.	311	
Total delta T, radiative plus convective, deg K	1222	
Total dT/alpha	873	

The above calculation shows a capillary length of 3.50 cm and a capillary diameter of .45 cm. The PFN impedance is 900 milliohms, and the capillary resistance is 735 milliohms, which is essentially a perfect match. This high impedance guarantees that less than 1% of the stored energy is lost in the 3 milliohm external resistance.

The pulse rate is pushed as high as possible to 208 pulses per second at 5 kW. The stored energy is then 24.0 J/pulse, the breakdown voltage is 2.45 kV, and the PFN capacitance is 8.00 microfarads. The pulse length is then 14.4 microseconds, which is longer than the characteristic venting time of 5.54 microseconds.

Instantaneous power is 1.64 MW at 1.14 eV (13,300 deg. K), and the hydrogen sound speed is, from the Sesame Tables [10], 11.9 km/sec at a specific heat ratio of 1.28. The average pressure is calculated from the propellant mass evaporation rate and choked conditions at the entrance to the supersonic nozzle to be 100 atm. This pressure is significantly above that of a DC arcjet, but is also significantly below the pressure capability of a capillary discharge from a mechanical standpoint.

The enthalpy from the charts (Fig. 6) [11] is 615 kJ/gm, which if fully expanded gives an ideal specific impulse of 3580 seconds, and an ideal thrust to power ratio of .057 N/kW. At 5 kW, the ideal thrust would then be 0.28 N.

The above LOTUS calculation also shows a calculation for insulator temperature rise, for an "alpha" factor of 1.4 for silicon nitride ceramic. The total calculated temperature rise is 1200 deg K, which requires an initial wall temperature of 700 deg K or less to stay below the Si_3N_4 melt temperature. This calculation is conservative, because the peak insulator temperature rise of 911 deg K does not occur simultaneously with the convective delta T of 311 deg K. The assumed pulse conditions therefore appear to provide a small margin against insulator ablation.

Frozen Flow Performance

The frozen flow performance is estimated from a simple model. Under discharge chamber conditions of 100 atm and 13,300 deg K the hydrogen is fully dissociated and 2% ionized (Fig. 6). We calculate the conditions for recombination of hydrogen atoms and then argue that since the resulting exhaust flow is mostly

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[10] SESAME Equation of State Library, Report LASL-79-62, Los Alamos Scientific Laboratory, Los Alamos, NM.

[11] "Mollier Chart for Hydrogen," AFOSR Contract AF 49(638)-54

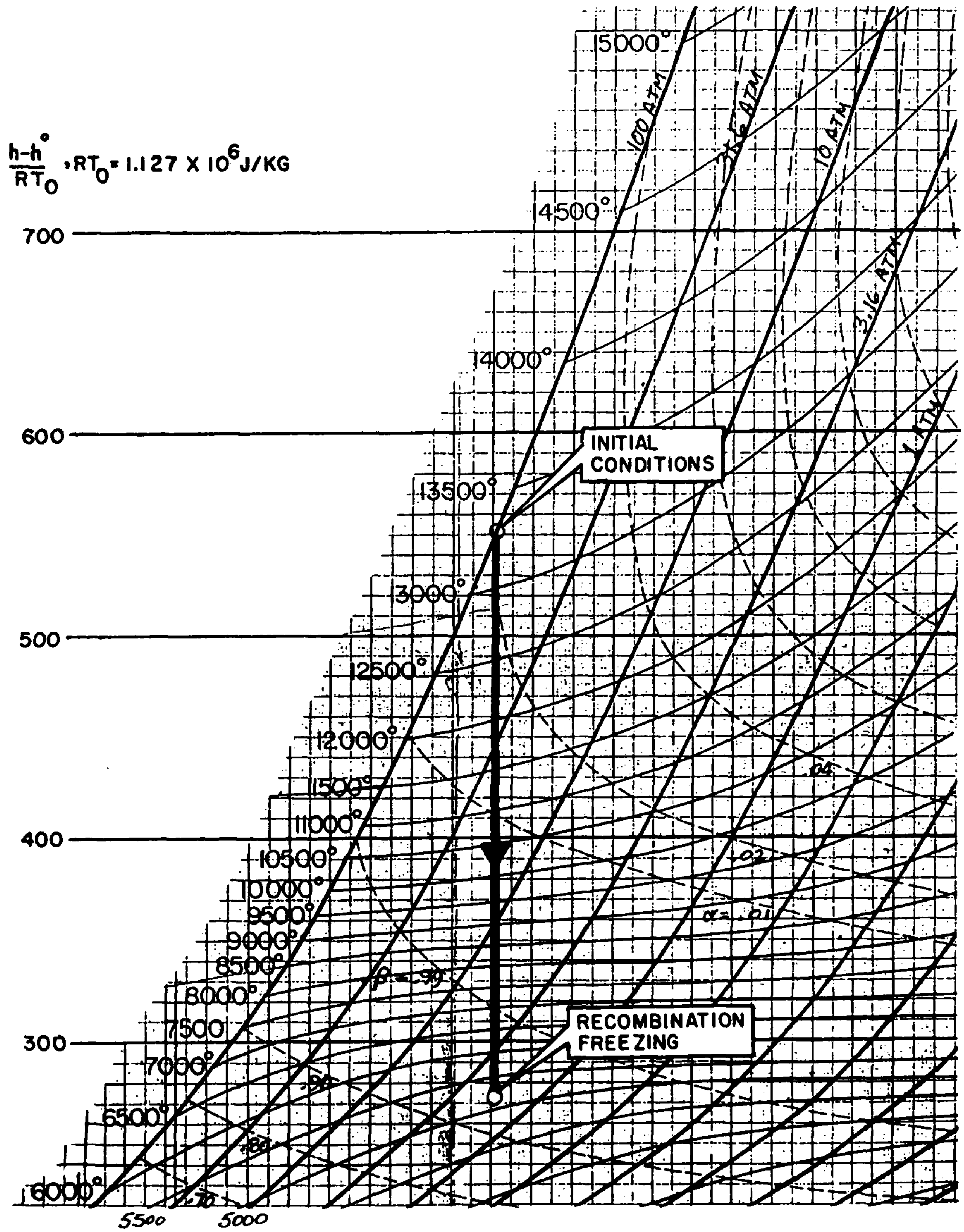

Figure 6. Mollier Chart for hydrogen, showing nozzle expansion process to point where recombination reactions freeze.

but not completely dissociated, the ionization energy in the exhaust is negligible compared to the dissociation energy.

The hydrogen starts from rest and accelerates to a maximum velocity of 25 km/sec. For a 10 cm long nozzle, a rough estimate of the nozzle residence time is $\approx$ 5 microseconds. From the model of Zel'dovich and Raizer [12], the recombination rate of atomic hydrogen is given by

$$\frac{1}{n_H}\frac{dn_H}{dt} \approx -\, 2\sigma\bar{v}\, V_{H_2}\, n_H^2$$

where

σ = scattering cross section $\approx 12 \times 10^{-16}$ cm^2

V_{H2} = volume of H_2 molecule $\approx 1.1 \times 10^{-23}$ cm^3

$\bar{v}$ = mean velocity, $(8kT/\pi M_H)^{1/2}$ = 1.56×10^6 cm/sec @ 1 eV

n_H = atomic density = 6×10^{19}/cm^3 @ 100 atm, 1 eV

This equation displays the important result that the percentage reduction in n_H scales as n_H^2. Operation of the PET thruster at high pressure therefore greatly aids the recombination process.

The characteristic recombination time from the above equation is

$$t_r = \frac{1}{2\sigma\bar{v}\, V_{H_2} n_H^2}$$

At 100 atm and 1 eV, $t_r = 6.7 \times 10^{-9}$ sec, which is so short compared to the flow time that the hydrogen is in local thermal equilibrium. The density at which t_r becomes longer than the 5 microsecond flow time is $n_H \approx 2.2 \times 10^{18}$, which corresponds to a pressure of about 1.8 atm and a temperature of about .44 eV (5200 deg K).

Under these conditions the hydrogen is about 93% dissociated (see Fig. 6), with an energy of 310 kJ and an enthalpy extraction of 305 kJ/gm. Were the nozzle to be truncated at this point, the efficiency would be 50% and the specific impulse would be 2520 seconds. We approximate the flow from this point as monatomic

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[12]Zel'dovich and Raizer, "Physics of Shock Waves and High-Temperature Hydrodynamic Phenomena," Vol I, Academic Press, NY, 1966, p.364 ff.

hydrogen, and calculate the density at which the hydrogen freezes in translation.

Translational freezing occurs when the mean free path becomes large, say 1 cm. This density is approximately $10^{15}/cm^3$, which for adiabatic flow of a perfect monatomic hydrogen gas with $\gamma=5/3$ corresponds to a drop in temperature from 5200 deg K to 30°K, which says that the translational energy does not freeze. [This is not true in practice because of the nozzle boundary layer.]

The design area ratio is 500:1. This ratio will inviscidly achieve Mach 20 with a monatomic gas, extracting over 99% of the translational enthalpy. This extracted enthalpy amounts to 107 kJ/gm, which when added to the initial 305 kJ/gm gives 412 kJ/gm extracted. The remaining 203 kJ/gm [615-412] is the dissociation energy of the hydrogen. The thruster efficiency is then

$$\eta = \frac{412}{615} = .67$$

The thruster specific impulse $\sqrt{2\Delta H}/g$ is

$$I_{sp} = 2930 \text{ seconds}$$

The thrust to power ratio $2\eta/gI_{sp}$ is

$$T/P = .0467 \text{ N/kW}$$

This design provides slightly more specific impulse and slightly less efficiency than was initially proposed, but provides a benchmark design against which 70-75% efficiency can be achieved at 2500 seconds.

Thruster Thermal Design

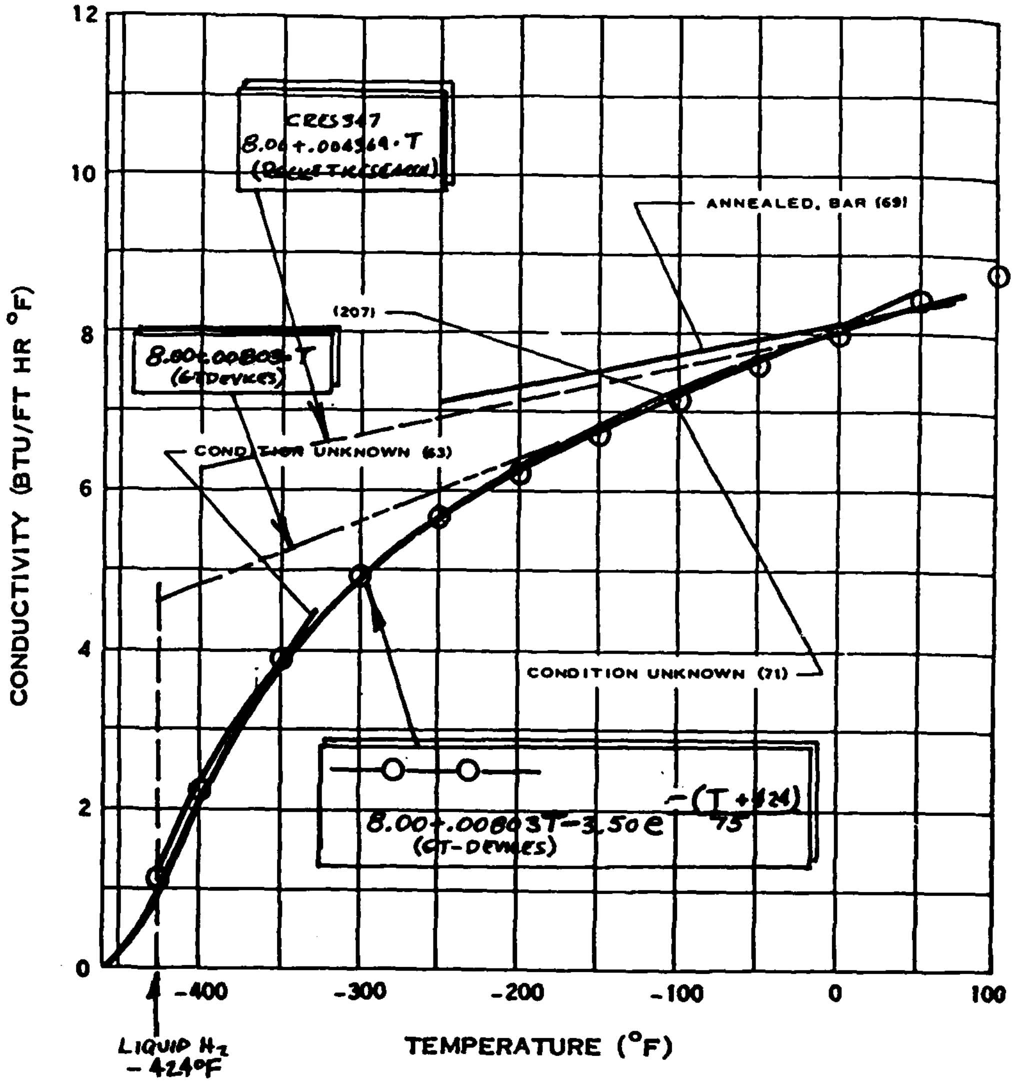
Rocket Research Company, Redmond, WA was hired to develop a liquid hydrogen PET thermal model. The principle concern was development of a design to prevent boiling of liquid hydrogen in the injection tube, and a satisfactory design was achieved.

The thruster design is shown in Figure 5. The primary components are listed in Table I below:

TABLE I. Liquid Hydrogen PET Thruster Components

No.	Components	Material	ODxIDxL, cm	Comments
1.	LH_2 Plenum	347 SS	1.1 x 0.8 x 5.0	
2.	Nozzle Seal	Carbon	2.3 x 0.5 x .10	
3.	Injection Tube	347 SS	0.5 x .45 x 8.0	k=.10 [W/cm°K]
4.	Injection Orifice	Macor	0.7 x .01 x 0.8	k=.017 [W/cm°K]
5.	Insulator	Si_3N_4	4.0 x .45 x 7.3	k=.25 [W/cm°K]
6.	Nozzle Electrode	W-alloy	2.3 x .50 x .95	k=.94 [W/cm°K]
7.	Nozzle	brass	8.5 x 1.3 x 10.4	A/A*=250
8.	Nozzle Mount	copper	6.1 x 3.0 x 1.5	k=3.8 [W/cm°K]
9.	Fin Radiator	copper	17 x 6.1 x 1.0	400 cm^2
10.	Compression Sleeve	4340 STL	6.1 x 4.0 x 7.8	k=.50 [W/cm°K]
11.	Retainer	Macor	6.1 x 1.7 x 1.5	k=.017 [W/cm°K]
12.	Injection Electrode	W-alloy	2.7 x 0.7 x 2.0	k=.94 [W/cm°K]
13.	Gas H_2 Plenum	347 SS	2.1 x 1.1 x 5.0	
14.	HV Seal	Macor	4.0 x 2.7 x 0.5	rated 5 kV
15.	Injection Tube Seal	Carbon	0.8 x 0.5 x 0.5	

B.7.v



THERMAL CONDUCTIVITY OF 347 STAINLESS STEEL

REF (207) (SEE ABOVE):

Y. S. Touloukian et al.: <u>THERMOPHYSICAL PROPERTIES RESEARCH CENTER DATA BOOK</u>, Vol 1, Ch. 1, June, 1966.

Figure 7

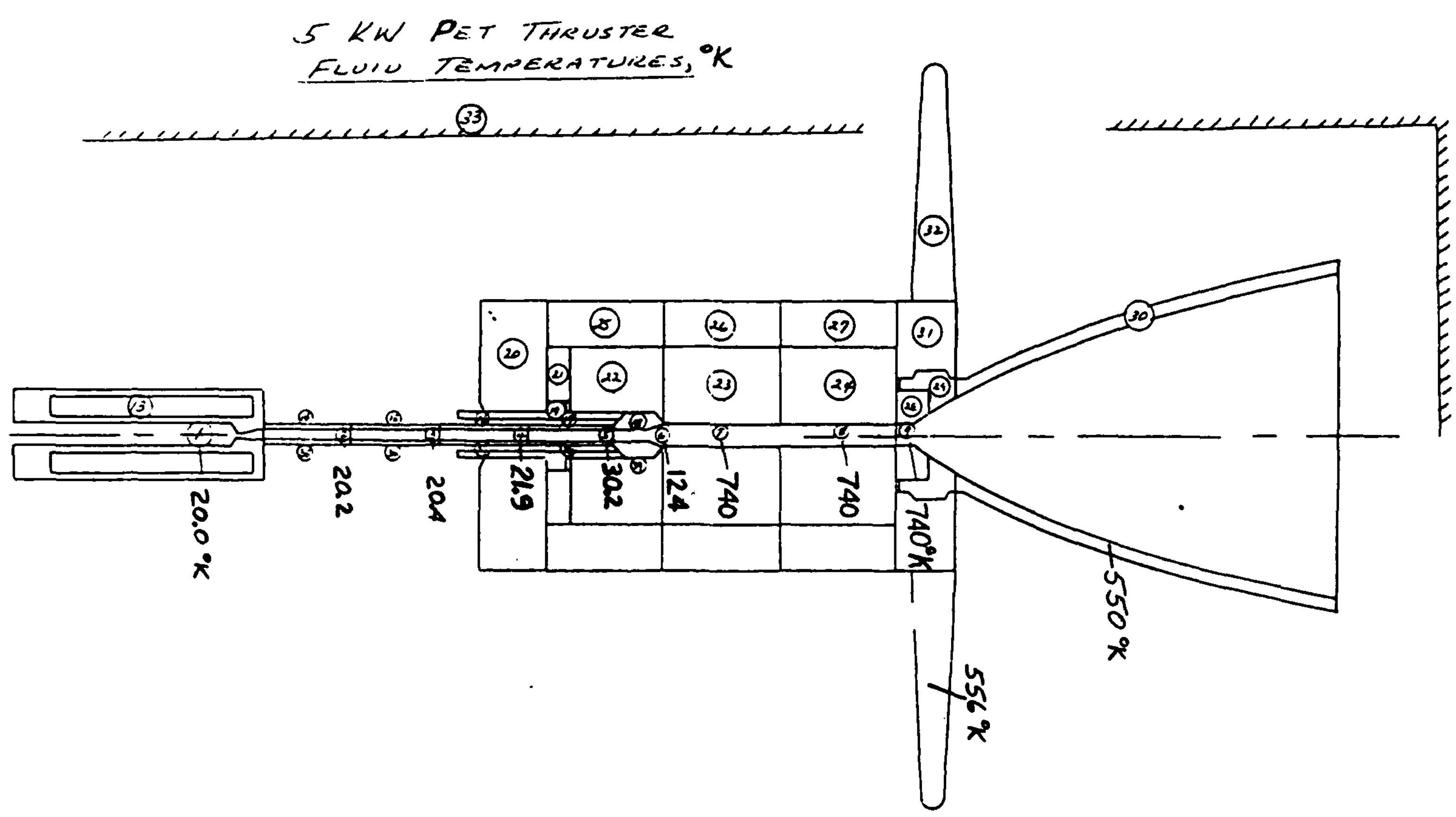


Figure 8. LH2 temperatures at 5 kW.

Thermal analysis was performed by a finite element radiation and conduction model. Heat is allowed to flow through material conduction paths between elements or by surface radiation to the surrounding environment, assumed to be at 293 deg K. Thermal conductivities were generally assumed to be constant, the lone exception being the injection tube, designed of 347 SS. For this material k drops by a factor 8 between room temperature and liquid hydrogen temperature of 20 deg K, so that k was expressed as an analytic function of temperature (Fig. 7)[13]. The elements used are shown in Figure 8.

Figure 8 also shows some of the principle calculated temperatures. The LH_2 is shown to start at 20.0 deg K, reaching 30.2 deg K before injection. Boiling can be prevented by maintaining the hydrogen at a pressure at 10 atm up to the injection orifice (Item 4, Fig. 5). Some of the hydrogen will then flash to vapor in the discharge capillary, but the effect of this on performance has not been analyzed.

The steady temperatures of the radiator (Item 9) and the nozzle (Item 7) were also calculated, and are shown in Figure 8. Based on these temperatures, total radiation to the environment is 365 watts, which represents a loss to the system which must be supplied electrically. This radiated loss has the effect of reducing the efficiency from .67 to .62.

The thermal analysis predicts that the 5 kW liquid hydrogen PET thruster would appear significantly different from an arcjet at the same power. Peak temperature of the radiator is only 550 deg. K, so that it would not radiate in the visible. This characteristic also makes the PET thruster hospitable to sensors located in the vicinity of the device.

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[13]"Cryogenic Materials Data Handbook," Vol. I, Sections A, B, C, Air Force Systems Command, Wright-Patterson Air Force Base, OH, NTIS AD713619. July, 1970.

III. FEASIBILITY OF LIQUID HYDROGEN LABORATORY TESTS

Although small quantities of liquid hydrogen are required for testing a 5 kW PET thruster, the preparation of a laboratory liquid hydrogen facility is a non-trivial task. At the design condition of .23 N thrust and 2930 seconds specific impulse, the mass flow rate of LH_2 is 8.0 milligrams/second, and the volume flow rate is .11 cm^3/sec. Several approaches were investigated for laboratory testing of PET with this LH_2 flow requirement.

Dewar System

The dewar system is the first choice, because it is the least expensive, and permits tests of any duration by replenishing the dewar during the test. The dewar system (Fig. 9) uses a liquid nitrogen pre-cooling system and a liquid hydrogen system, which consists of a dewar, transfer line and phase separator. [The LN_2 system is not shown in Fig. 9] The phase separator is located close to the thruster, and uses gravity to insure that only liquid phase hydrogen enters the thruster plenum (Item 1, Fig. 5). The separated gas is ducted to a cooling jacket surrounding the plenum (Item 13, Fig. 5) and is then ejected into the laboratory safety system.

In order to cool the system initially, the system mass is first cooled to 77°K, using LN_2. The LH_2 is used to cool down to 20°K. The system mass is not known exactly, but it will be in the 5-10 kg range. A rough estimate of the quantity of LH_2 required per kg for cooling is derived for an average specific heat of 300 J/kg-deg K, a temperature drop of 57 deg K, and a heat of evaporation of 31 kJ/liter. Thus .55 liters of LH_2 are required for cool-down, so 3-6 liters are required for a 5-10 kg system, an acceptable amount.

Once cooled, LH_2 is required for thruster operation. The LH_2 quantity for 5 kW thruster operation is .11 cm^3/sec or 0.4 liters/hour. Since the thruster takes 1.4 hours for 10^6 pulses, relatively little LH_2 is required to run the thruster, except for the longest tests. The longest test requirement would probably be 10^8 pulses, which would require 140 hours and 56 liters, easily tolerated by a dewar system. Even a 50 kW thruster could be tested by this technique, since a standard pressurized LH_2 cylinder holds 170 liters, and can be refilled during a test.

Liquid hydrogen is transferred from the dewar under pressure to the thruster through a vacuum-jacketed transfer line. The design of a typical line for helium [and hydrogen] is shown in Figure 10, as available from CRYO Industries of America, Inc.

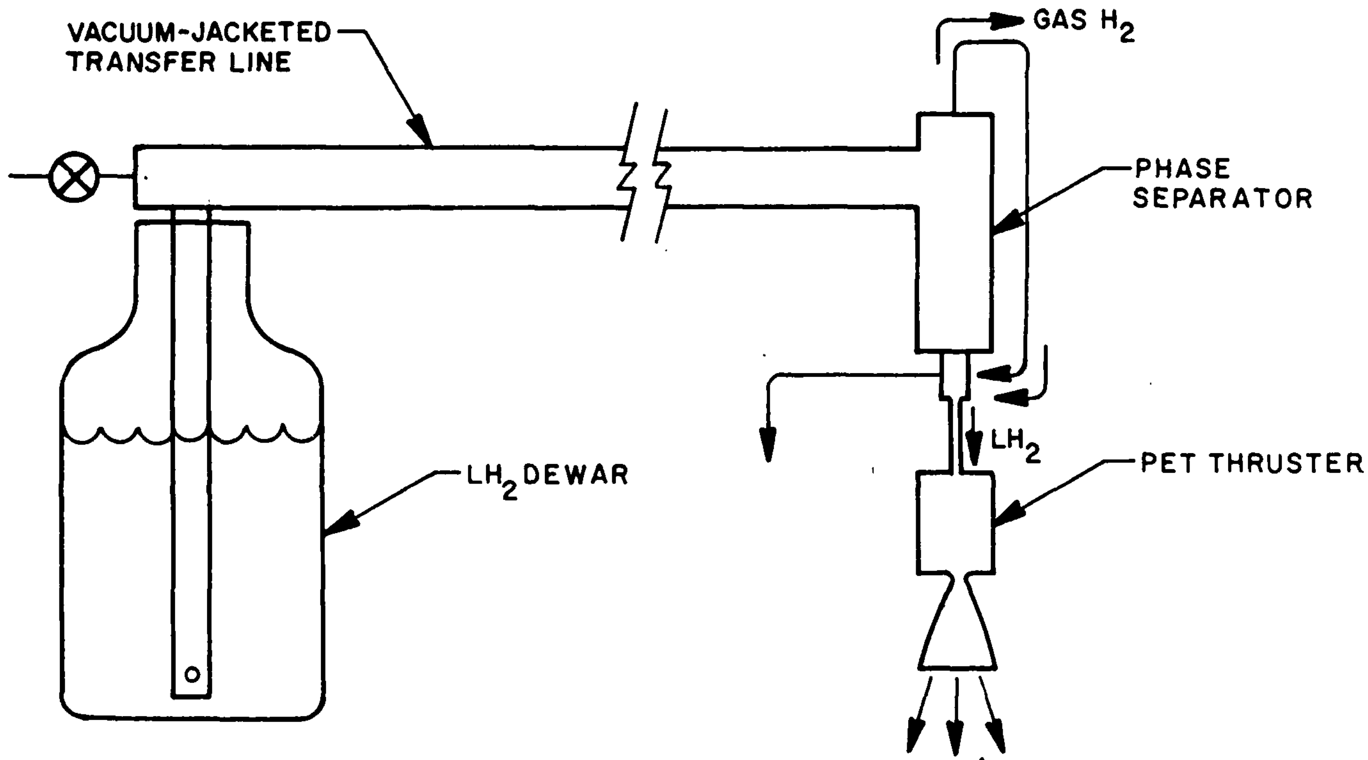


Figure 9. Dewar System for PET Thruster testing with LH_2.

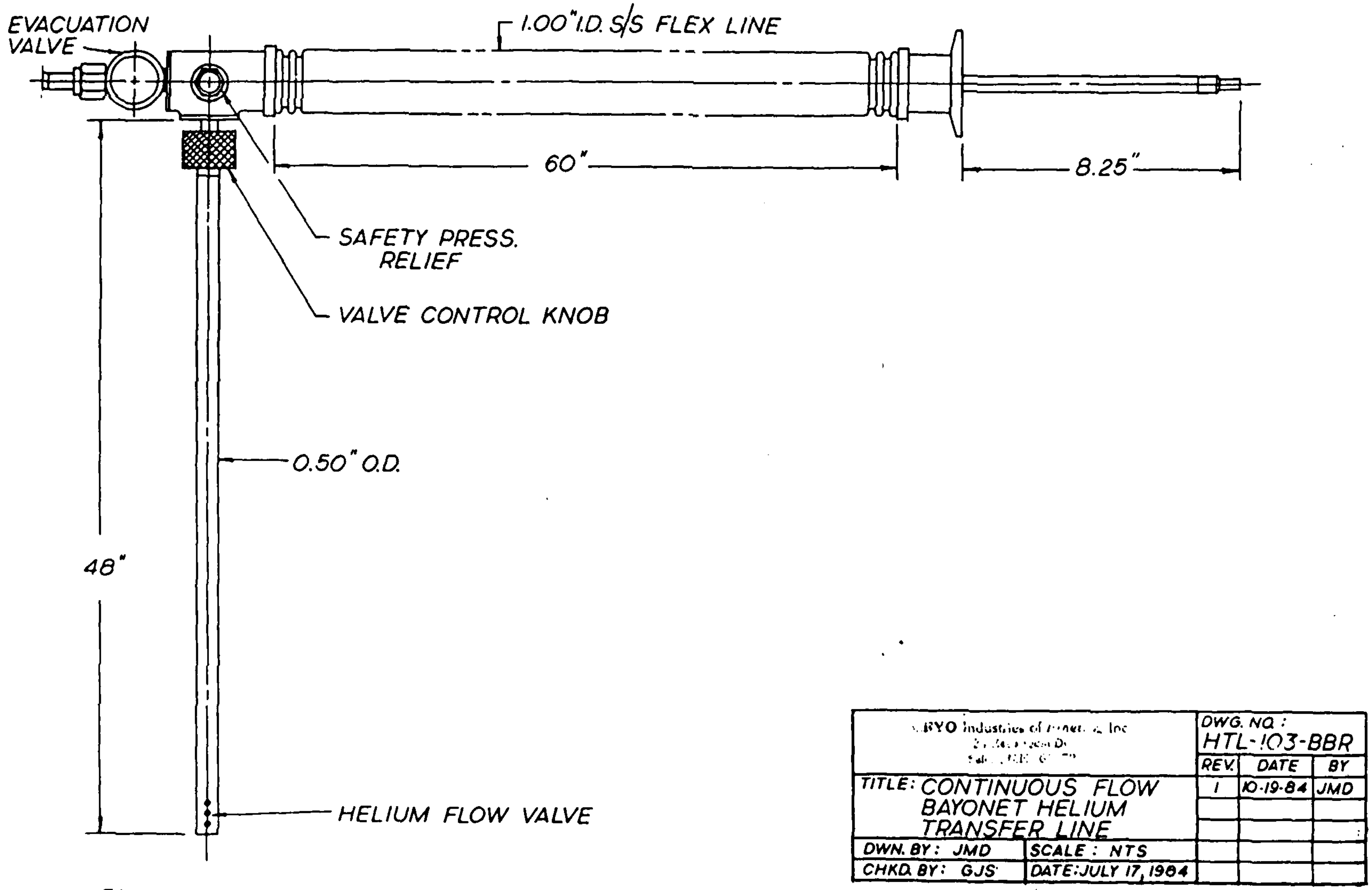


Figure 10. CRYO Industries cryogenic transfer line.

The phase separator (Fig. 9) uses gravity to extract hydrogen vapor from the fluid, which flows to the thruster, shown with thrust axis vertical. The cold vapor is ducted to a plenum jacket surrounding the LH_2 plenum. Additional thermal insulation in the forum of radiation shielding and jacketing will be required, but is not included in this design.

The most difficult requirement to satisfy on the liquid hydrogen system is safety. On large LH_2 systems, no valves are allowed inside a building for safety reasons, which requires that 100 m or so of transfer line be cooled. The problems of greatest concern are:

a) Accidental spillage and fire
b) Leakage of pressurized hydrogen and fire
c) Purging and disposition of vacuum pump exhaust
d) Hoods and ventilation system for hydrogen gas
e) Accidental spillage on human skin.

The solution of each of these areas is possible, and requires great care and experience with cryogenics and in particular with LH_2. No attempt was made to estimate safety-related costs, but they are certainly significant.

Refrigerator System

The refrigerator system, in which LH_2 is made at the thruster from gaseous hydrogen, is significantly safer than the dewar system, because the problem of LH_2 spillage does not exist. The basic system is shown in Figure 11.

For hydrogen, the heat which must be extracted to condense from 77 deg K is 1200 J/gm. For a 5 kW thruster, the equivalent refrigeration power is 9.8 W. This can be supplied by a small closed cycle refrigerator, e.g. the Air Products CS-208L shown in Figure 12. This refrigerator is too small, however, to provide the cool-down requirements of the system. A 10 kg system would require approximately 160 kJ, so a two hour cool-down would require an average of 22 watts. Even though the refrigerator efficiency and power is higher at 77 deg. K a larger refrigerator than the CS-208L would be required.

A solution to this problem is to provide a large (say 100 kg) block of copper, which is cooled to the LH_2 freezing point of 14 deg. K. The total heat capacity of the block from 14 deg K to 20 deg K is then 180 kJ, which can provide the 160 kJ required for cool-down. A small 10 W refrigerator can then be used, running continuously. Note that a typical cost for a 10 W refrigerator is \$16k, whereas a 25 W refrigerator is about \$25k.

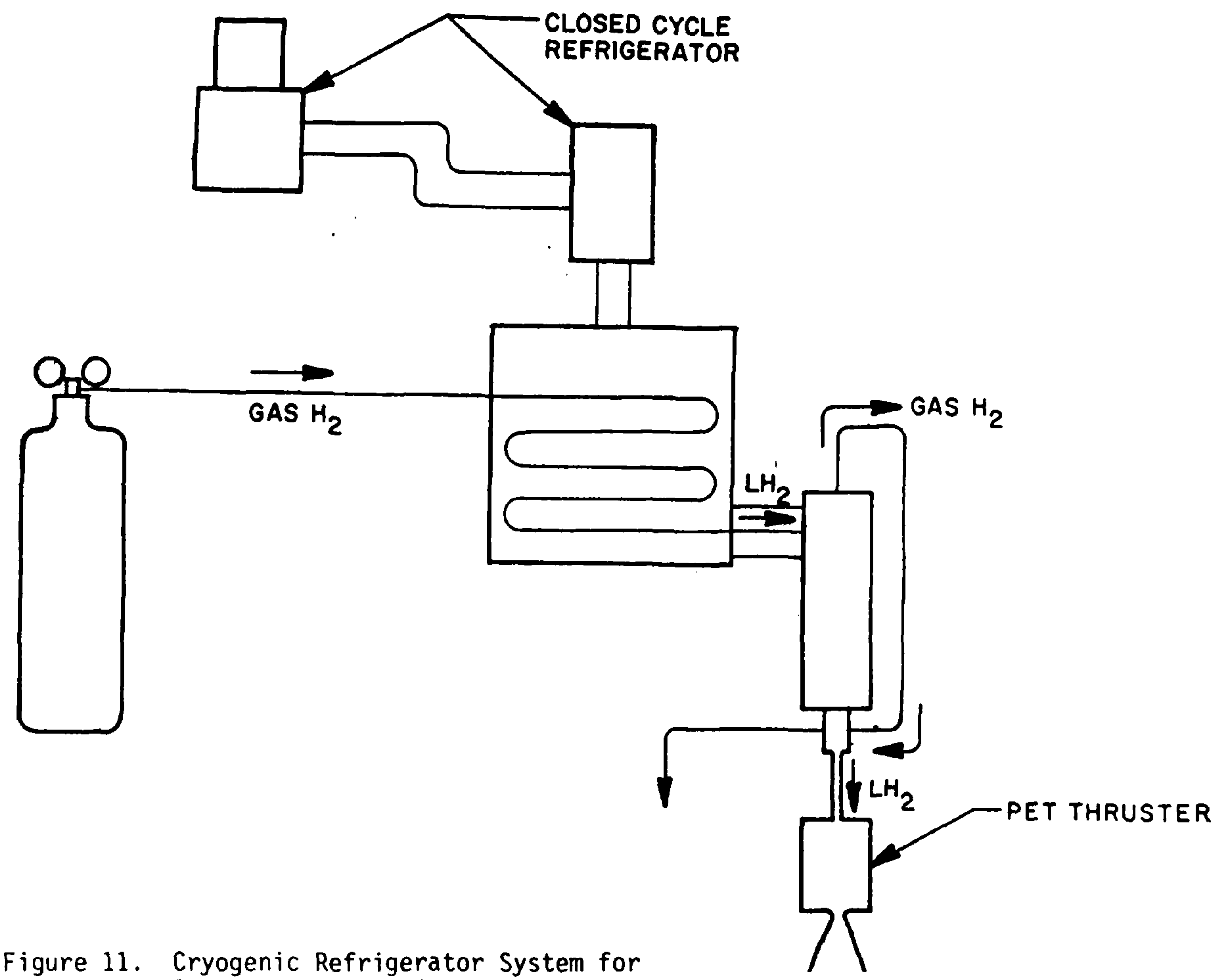
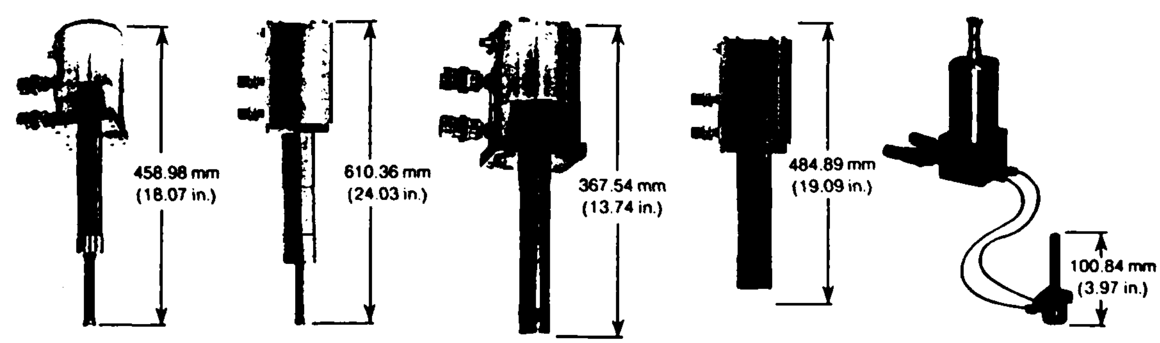


Figure 11. Cryogenic Refrigerator System for PET THruster testing with LH_2.

DISPLEX® CLOSED CYCLE REFRIGERATION SYSTEMS

7

The Displex® system has been designed to meet the demands of laboratory research and industrial cooling applications. The design objective has been to provide *Low Cost, Reliable, Field Maintainable* cryogenic refrigeration. The following table outlines the various models of the product line and their capabilities.



COLD END CS-202 | CS-208L CS-208R CS-204 | CS-102 | CS-108 CS-104 | CS-1003

Although the refrigerator system is safer than the dewar system, there are still important safety considerations related to gaseous hydrogen. These safety issues, items b-d above. require design and modification to any laboratory not already equipped to handle toxic or combustible gases.

Existing Facilities for LH_2

Although technically feasible, the time and cost for designing, procuring and learning to operate a new liquid hydrogen facility are large for an SBIR Phase II program. An alternative is to use an existing facility, which could provide not only the LH_2, but also the vacuum tanks and equipment required for PET thruster testing.

One such facility is the 10V Aerospace Chamber at the Arnold Engineering Development Center, Arnold Air Force Station, TN [14]. The description of the 10V facility is reproduced in Appendix A-4 of this report, which shows that 5 kW PET thruster testing can be performed in this facility.

A visit was made [5/27/87] to AEDC to see the 10V facility and discuss LH_2 operation with Dr. Allan B. Bailey of Calspan. PET requirements were thoroughly discussed, and there is confidence that an experiment could be performed at AEDC on the 10V. Operating at 10 atm pressure could prove to be a problem, however. Besides LH_2, LN_2 and LHe are available. AEDC has expertise in transfer line fabrication, LH_2 handling, and also in the testing of small rocket thrusters using a variety of diagnostics. AEDC has made an ROM estimate of $12-$22 K as the cost of safety analysis plus hardware to permit LH_2 operation. Other costs would include a power supply, a tank cooling system, and GT-Devices travel costs for testing, in addition to AEDC testing fees.

Phase II R&D for PET Thruster Scientific and Technical Feasibility

The most critical issue facing the liquid hydrogen PET thruster is the LH_2 injection and arc breakdown process. As discussed above, hydrogen boiling must be prevented in the injection tube, which can be prevented by proper design, particularly the use of a ceramic injection nozzle. This design must be verified by experiment. The second major issue is the evaporation of the injected liquid in the discharge chamber, which must be accomplished on a microsecond time scale.

---

[14] "Test Facilities Handbook" (Twelfth Edition), Arnold Engineering Development Center, Arnold Air Force Station, TN, March 1984.

The details of the liquid/vapor heating process for LH_2 subjected to a megawatt-level electric discharge depend on the injector design. Drop size and wall interactions will strongly influence the evaporation time. This critical area is currently being investigated at GT-Devices under an AFOSR-funded 36 month experimental program to investigate the arc heating of water sprays spectroscopically. The results of this investigation will aid in understanding of the LH_2 injection process.

One of the related questions to the injection process is that of propellant utilization, i.e., how much propellant must actually be injected to hold the discharge temperature at the design level. Any excess propellant has the effect of reducing specific impulse. While it is believed that high (over 80%) utilization factors can be achieved, this remains to be demonstrated.

Chemical reactions between hydrogen and the thruster electrodes are an important issue. Total pulse time for a long (10^8 pulses) mission is approximately 10^3 seconds, so it is hoped that chemical reactions can be minimized by the proper choice of materials. The solution to this problem must be demonstrated by experiment.

Finally, the performance (thrust and specific impulse) must be measured. While the calculations of the heating and exhaust processes provided in this report are believed to be a fair representation of the actual situation, the issue can only be resolved by experimental test.

Summary of LH_2 Feasibility

Three approaches were investigated; dewar, closed cycle refrigerator and the AEDC 10V aerospace chamber. Startup costs and startup time are lowest for the AEDC option. This option requires GT-Devices offsite personnel, but this would be a requirement in any case unless high vacuum facilities were constructed at GT-Devices. Testing at AEDC therefore seems to be the best solution investigated.

The refrigerator option should be rejected. It suffers from high equipment costs, which are not offset by the savings in safety requirements. The flexibility and low initial cost of the dewar system makes this system the second choice after AEDC.

IV. CONCLUSIONS RELEVANT TO A PHASE II SBIR PROGRAM

The purpose of the Phase I program was to design a liquid hydrogen PET thruster and determine the feasibility of liquid hydrogen testing for Phase II.

The design analysis indicates that the LH_2 PET would be a 62% efficient thruster at 2930 seconds, performance significantly higher than other types of thrusters (Fig. 12). Operation at higher pressures and lower temperatures, if it can be achieved, would raise the system efficiency to the 75% level at 2000 seconds.

The current history of space operations strongly suggests that propellants other than liquid hydrogen, particularly water and hydrazine, will have a role to play in space before LH_2 becomes available. As far as the PET thruster is concerned, it may be well to establish water and hydrazine performance before liquid hydrogen, since the former can still perform in an interesting specific range above 1000 seconds.

Performance measurements on PET will require testing at a laboratory facility capable of high vacuum not available at GT-Devices. Initial testing on water can be performed at GT-Devices inexpensively at modest tank pressures.

It is concluded that Phase II of this PET performance effort should be broadened to include water, hydrazine and liquid hydrogen propellants, and this has been proposed as indicated in Appendices A-1 and A-2.

GTD

ELECTRIC PROPULSION PERFORMANCE

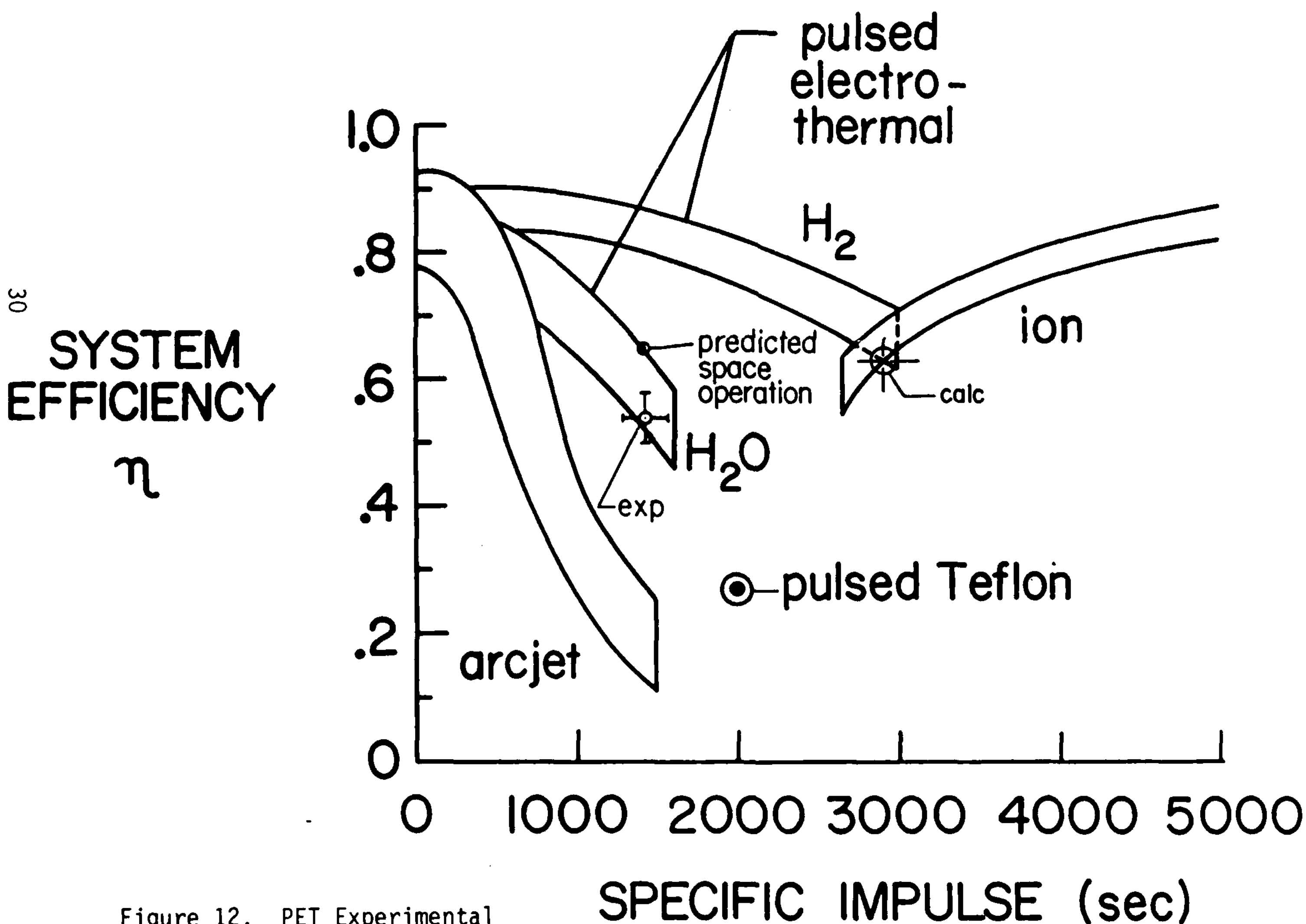


Figure 12. PET Experimental and Calculated Performance for LH_2.

APPENDIX A - 1

GT - Devices

5705A General Washington Drive
Alexandria, VA 22312
(703) 642-8150

11 September 1987

Ms. Linda Marie Kendrick
Contract Specialist
3350 (LMK)
NASA Lewis Research Center
Cleveland, Ohio 44135

Dear Ms. Kendrick:

GT-Devices is pleased to submit the enclosed SDIO SBIR Phase II proposal GTD-93, "Performance of a Pulsed Electrothermal Thruster with Water, Hydrazine and Liquid Hydrogen."

Phase I of this effort was performed under AFOSR Contract F49620-87-C-0028DEF. After Phase I, Leonard H. Caveny of SDIO/IST agreed to assign responsibility for technical evaluation of this Phase II proposal to NASA-Lewis, under the direction of David C. Byers, Chief, Auxiliary, Propulsion Branch.

We are aware that the SBIR program gives special consideration to proposals demonstrating non-Federal capital commitments for Phase III activities. We are pleased to say that Rocket Research Company, Redmond, WA, has made such a commitment for this Phase II effort, and has promised a growing commitment based on successful results in Phase II. This commitment takes the form of 1) equipment purchases 2) facilities modifications, and 3) the performance of Task 4 of this proposal (fuel cart) at no cost to the contract. Rocket Research Co. has agreed to do this in order to stay within the SBIR funding eligibility requirements, and to demonstrate their company's commitment to this vital program. Rocket Research Company's letter of commitment is enclosed in this proposal.

If you have any questions of a technical nature, please call Rodney L. Burton at the above number. Other questions may be referred to me.

Sincerely,

Marian E Ludlow

Marian E. Ludlow
Contracts Administrator

Enclosure

cc: R. L. Burton, D. A. Tidman, S. A. Goldstein
D. C. Byers - NASA-Lewis
L. H. Caveny - SDIO/IST
W. W. Smith, R. J. Cassady, B. W. Schmitz - RRC

APPENDIX A - 2

11441 WILLOWS RD N.E.
P.O. BOX 97009
REDMOND, WA 98073-9709
(206) 885-5000 TWX 910-449-2861

Olin ROCKET RESEARCH COMPANY
DEFENSE SYSTEMS GROUP

September 9, 1987
Ref. No. 3100-87-169

Dr. R. L. Burton
Director of Space Applications
GT Devices
5705A General Washington Drive
Alexandria, VA 22312

Dear Dr. Burton,

Rocket Research Company (RRC) is pleased to have the opportunity to cooperatively pursue the development of the Pulsed Electrothermal Thruster (PET) with GT Devices. The PET performance regime and adaptability to many propellants compliments the range of electric thrusters currently being pursued by RRC. This SBIR provides the opportunity to prove performance feasibility of this important new thruster concept and as such has generated a company commitment to support this effort.

RRC is modifying an on-going electric propulsion capital improvement effort to accommodate the particular testing requirements of the PET. As part of this capital improvement, RRC will provide a fuel cart capable of running N_2H_4 or H_2O, will modify a new vacuum chamber to allow vertical firing of the PET, and will provide adequate instrumentation, data recording, power, and pumping capacity for the proposed PET test. If the proposed feasibility test is successful and applications for this thruster become more defined, RRC's commitment in this area will continue to grow. Again, RRC is pleased to work with GT Devices and focus its experience base and talent on a successful Phase II effort.

Sincerely yours,

Bruce W. Schmitz

B. W. Schmitz
Vice President, Engineering and Technology

BWS/km#54

APPENDIX A - 3

Liquid Hydrogen Experimental Project for G T Devices Inc.

April 20, 1987

Project Director: Dr. Rodney L. Burton
Director of Experimental Research

Purpose:
To determine the feasibility of suppling Hydrogen as a liquid fuel to a experimental booster motor at a rate of 1 Gram per minute.

Physical layout:
The motor will be placed on a vacuum chamber in a way so as to allow the discharge from the motor to enter the chamber. This is to simulate operation in a vacuum such as outer space.

The fuel will be supplied from a inlet at the top of the motor which will be in a vertical position on the top of the vacuum chamber, and this fuel must be a liquid saturated a 1 pound per square inch gauge.

DATA RELATED TO HYDROGEN

Conversion data:

	weight		gas		liquid	
	Lb.	Kg	SCF	Nm3	Gal	L
1 pound	1.0	0.4536	192	5.047	1.6928	6.408
1 kilogram	2.205	1.0	423.3	11.126	3.733	14.128
1 SCF Gas	0.005209	0.002363	1.0	0.02628	0.00882	0.03339
1 Nm3 Gas	0.19815	0.08988	38.04	1.0	0.3355	1.2699
1 Gal liq.	0.5906	0.2679	113.41	2.981	1.0	3.785
1 L liq.	0.15604	0.07078	29.99	0.7881	0.2642	1.0

Notes: standard cubic foot gas measured at 1 atomosphere and 70'F.

Nm3 gas measured at 1 atomsphere and 0'C.

Liquid measured at 1 atmosphere and boiling temperature.

Hydrogen gas values expressed in the stable condition 75% ortho, 25% para.

Equilibrium Hydrogen is commonly referred to as para-Hydrogen and is 2% ortho-Hydrogen.

Hydrogen liquid values expressed in the stable para condition.

All values rounded to nearest 4/5 significant numbers.

All values are consistent with standards adopted by the Compressed Gas Association on June 19.1962.

The following is the Hydrogen material data for refrigerated liquid which is synonymous with the terms liquid Hydrogen, Para Hydrogen, and liquified Hydrogen.

D.O.T. identity number; 1333-74-0

D.O.T. hazard class: FLAMMABLE GAS

Chemical family: Cryogenic Flamable inorganic

Volumetrics: One volume of liquid Hydrogen at its boiling point and atomspheric pressure will vaporize into approximately 850 volumes of gaseous Hydrogen at 70'F (21.1'C) and 1 atmosphere.

HEALTH HAZARD DATA:
Hydrogen is defined as a simple asphyxiant. Oxygen levels should be maintained at a greater than 18 Molar percent at normal atmospheric pressure which is equivalent to a partial pressure of 135 MM Hg. (ACGIH,1985-86). Inhalation of high concentrations of Hydrogen so as to exclude an adequate supply of oxygen to the lungs causes dizziness, deeper breathing due to air hunger causes possible nausea and eventual unconsciousness. Contact with the cryogenic liquid can cause tissue freezing or frostbite on dermal contact or if splashed in the eyes. Hydrogen is inactive biologically and essentially nontoxic.

Prompt medical attention is mandatory in all cases of overexposure to Hydrogen. Rescue personnel should be equipped with self-contained breathing apparatus and be cognizant of exteme fire and explosion hazard.

PHYSICAL DATA:
Hydrogen is flammable over a very wide range in air and should be considered a hazardous mixture when other liquids, solids, or gases are introduced.

BOILING POINT: -423'F (-252.8'C)

VAPOR PRESSURE: @70'F(21.1'C): above the critical temp. of -399.8'F

LIQUID DENSITY AT BOILING POINT: 4.43 Lb/Ft3 (70.96Kg/M3)

GAS DENSITY AT 70'F AND 1 ATM: .0052

SOLUBILITY IN H2O: very slight

EVAPORATION RATE: 99.9+% volatile

FREEZING POINT: -434.6'F (-259.2'C)

SPECIFIC GRAVITY: (AIR=1) @70'F (21.1'C)= .069

APPERANCE: Clear, colorless, odorless liquid

AUTO IGNITION TEMP.: 1058'F (570'C)

FLAMMABLE LIMITS % X VOLUME: LEL 4 UEL 74.5

NOTE: Earth-ground and bond all lines and equipment associated with the Hydrogen system. All electrical equipment should be non-sparking or explosion proof. Liquid Hydrogen cylinders should be used only in well ventilated areas of non-combustible construction and in accordance with the manufacturer's instructions. This type cylinder must always be kept in an upright position. Specialized hand trucks are needed for movement of this type cylinder.

The feasibility of this project is possible using the above stated safety and data guide lines.

The supply control for the liquid Hydrogen can be operated from a temp. sensor type control or a liquid flow control. Either type will supply the quality liquid required. This will be used along with vacuum insulated piping, special controls and equipment required to supply liquid to the motor. Serious considerations must be given to the safety of this project as it relates to the test area, supply of liquid Hydrogen and Oxygen sources. We are sure that some type of modification must be done in the test area to provide adequate ventilation due to the nature of the liquids and gases and their wide range of flammability.

With out additional data we can not expand on the physical layout which would explain the detail of the liquid Hydrogen installation with related equipment. There are two ways to supply the liquid to this project, and we feel that the smallest amount of liquid Hydrogen on test site would be the safest. To supply the test stand we would suggest that a small dewar be used to supply the phase seperator which in turn supply the inlet to the motor and provide a liquid source saturated at one atmosphere. Care must be taken to properly vent the phase seperator as a volume of Hydrogen gas must be vented at this point to assure the liquid is at the quality required. The dewar can be removed and refilled from a liquid source such as a liquid cylinder. This refilling process must be done at a remote and safe site. The liquid cylinder contains approx. 45 gallons of liquid Hydrogen which could be pressurized as high as 150 P.S.I.G. and adequate ventilation is required for the refill process.

The basic differance between a liquid dewar and a liquid cylinder is in the design and coding of the units. The dewar is an insulated container designed to hold liquids for a short period of time at atmospheric pressures and cryogenic temperature. The liquid cylinder is A.S.M.E. coded at a pressure and designed to hold larger amounts of cryogenic liquid under pressures higher than atmospheric for longer periods of time. The condition of the liquid will change as the pressure rises in the liquid container.

The control for the liquid Hydrogen must be done after the phase seperator as the volumetrics chang(. I would suggest that a control flow orfice be used to allow the liquid flow to be changed as desired during testing the motor. The control orfice can be designed several ways depending on the accuracy and response required. One possible way to build this control is to design a fixed orfice with a metering valve up stream which will allow the flow to the orfice to be adjusted. The entire liquid flow operation can be monitored through liquid pressure and temperature sensors which can be manually or automatically controlled.

The general layout and operation of the liquid Hydrogen supply will be done as follows.

1. The dewar will be filled from a larger liquid source such as a liquid cylinder.

2. The dewar will then be positioned on the test stand and a supply of Hydrogen gas will be applied to the unit which will cause the liquid to be forced at low pressure into the phase seperator.

3. The phase seperator will reduce the heat from the liquid and allow the cooled liquid to flow into a small reserve which will feed the motor. At this point the flow control equipment will be installed to adjust the flow.

4. the entire Hydrogen supply system should be shrouded in a Nitrogen gas blanket. This will assure that the supply sustem will remain inert and free from Oxygen impingement. The Nitrogen blanket will improve the safety of the system.

The cost involved with the liquid Hydrogen supply is difficult to assess but a qualified estimation of the total cost would be in the $35,000.00 range. This estimate is with the understanding that all the equipment is of a prototype design that requires special handling.

In closing, any experimental project involving liquid Hydrogen can be dangerous if not properly controlled. This project is feasible only if the safety control is strictly adhered to and the liquid volumes are maintained at the lowest possible levels.

Thank you for the interest in Woodland Cryogenics, our products, and services. We look foward to the continuation of this and other projects in which we can provide this type of service.

Yours Truly

Hudson Payne
Sales Mgr.
Woodland Crogenics Inc.

APPENDIX A - 4

PART 10
Aerospace Chamber (10V)

10.1 GENERAL DESCRIPTION

The Aerospace Chamber (10V) (Fig. 10.1) is designed for low density aerodynamic studies and testing of small rocket engines under space vacuum conditions.

Fig. 10.1 View of the Aerospace Chamber (10V)

The stainless steel chamber, located in the Developmental Chamber Area in the Engineering Laboratory Building, is 10 ft in diameter and 20 ft long. Chamber pumpdown is accomplished with a 730-cfm roughing pump, a 140-cfm forepump, and a 6-in. diffusion pump. The chamber is equipped with an internal cryogenic system designed for high efficiency vacuum pumping of aerodynamic test streams and rocket exhaust products. The cryogenic system consists of a 77°K liquid-nitrogen-cooled high capture liner, 20°K gaseous-helium-cooled surfaces and 4.2°K liquid-helium-cooled surfaces.

The unique capabilities of this chamber make it suitable for (1) aerodynamic testing of items, such as drag measurements on satellite configurations and high altitude air sampling probes, in the transition and free molecular flow regimes, and (2) determining the performance and effects of small rocket engines operaing in the space environment, including jet plume studies.

10.2 PERFORMANCE

10.2.1 AERODYNAMIC TESTING

The low denisty aerodynamic testing configuration is shown in Fig. 10.2. The nozzles are cooled with liquid nitrogen to reduce boundary layer thickness. A resistance heater allows operation at reservoir temperatures up to 1000°K at Mach numbers from 2 to 7.

Typical performance characteristics of this chamber as an aerodynamic facility are shown in Figs. 10.3a and b. Uniform flow core diameters up to 50 cm can be obtained. The run time, using nitrogen varies from a few minutes at a mass flow of 40 gm/sec to hours at mass flows less than 3 gm/sec. Molecular mean free path lengths of slightly more than 2 in. can be obtained at the nozzle exit plane.

10.2.2 SPACE PROPULSION TESTING

The space propulsion testing configuration is shown in Fig. 10.4. Free expansion on the jet plume is made possible by the high capture rate of the exhaust gases on the chamber cryogenic surfaces.

The performance of the chamber for rocket engine testing is shown in Fig. 10.5. The actual test conditions which can be achieved within the performance range shown are a function of the propellants used and the resultant exhaust products—the lower the hydrogen content, the lower the chamber ambient pressure.

Nominal Mach Number	Exit Diam, in.	Throat Diam, in.	Cone Half Angle, deg
2	9.85	7.33	Contoured
3	30.00	10.66	10.0
6	42.80	3.08	20.0

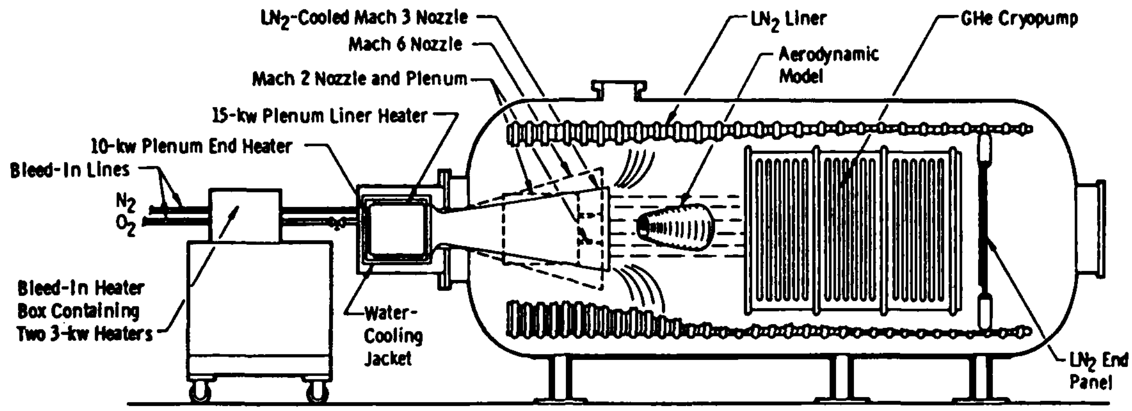
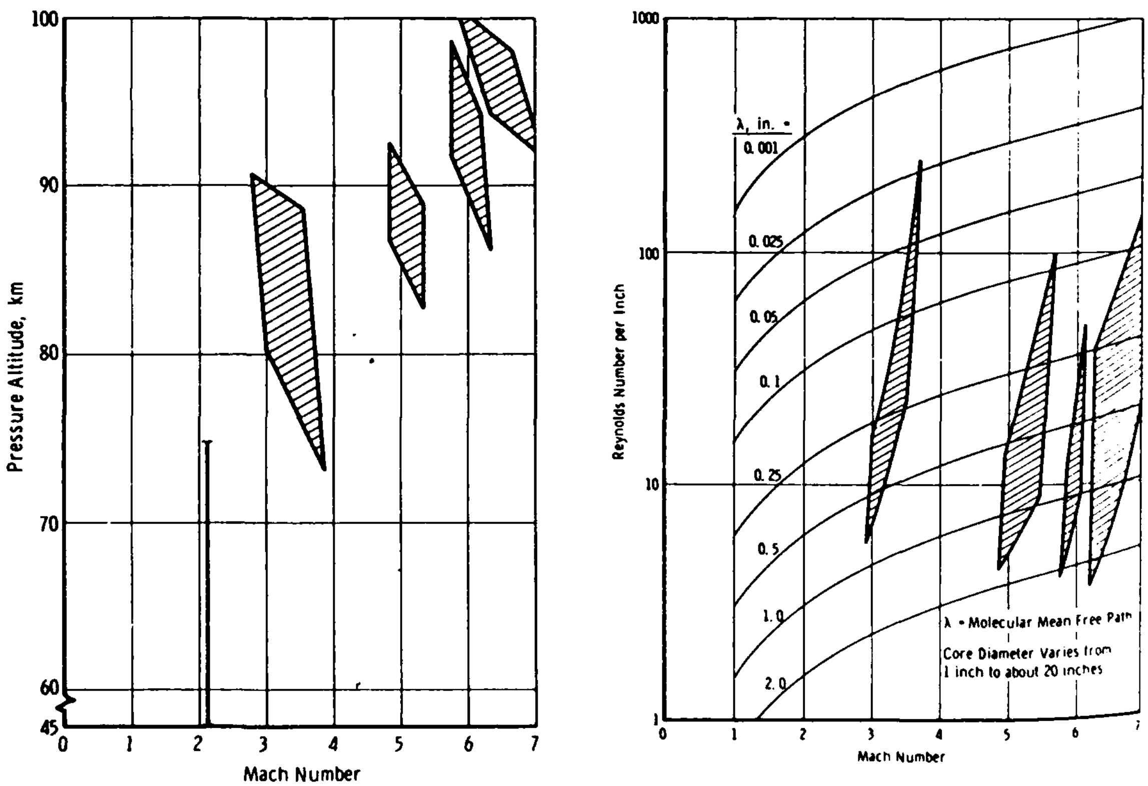


Fig. 10.2 Aerospace Chamber (10V) Operating as Low Density Aerodynamic Facility



a. Pressure Altitude versus Mach Number **b. Reynolds Number versus Mach Number**

Fig. 10.3 Performance of Aerospace Chamber (10V) for Aerodynamic Testing

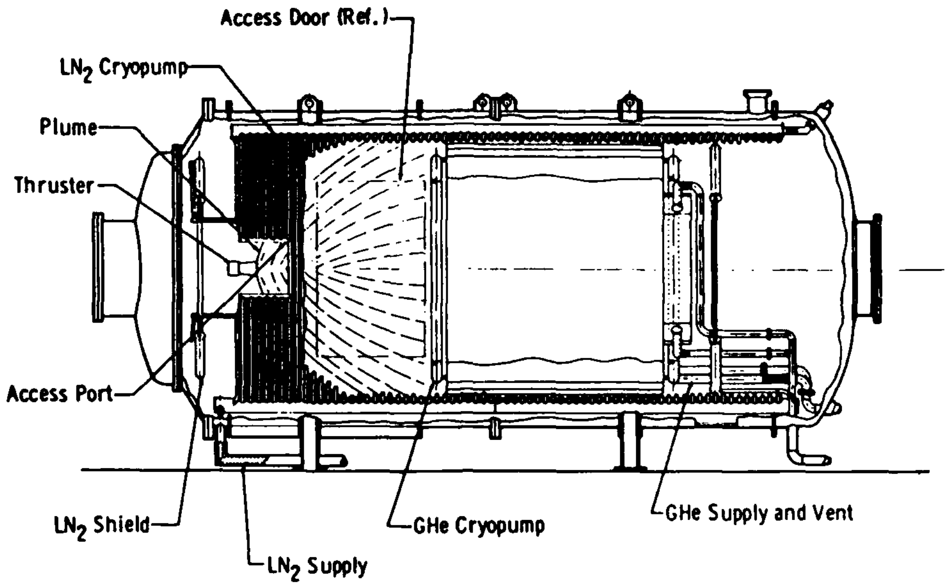


Fig. 10.4 Aerospace Chamber (10V) Operating as a Space Propulsion Test Facility

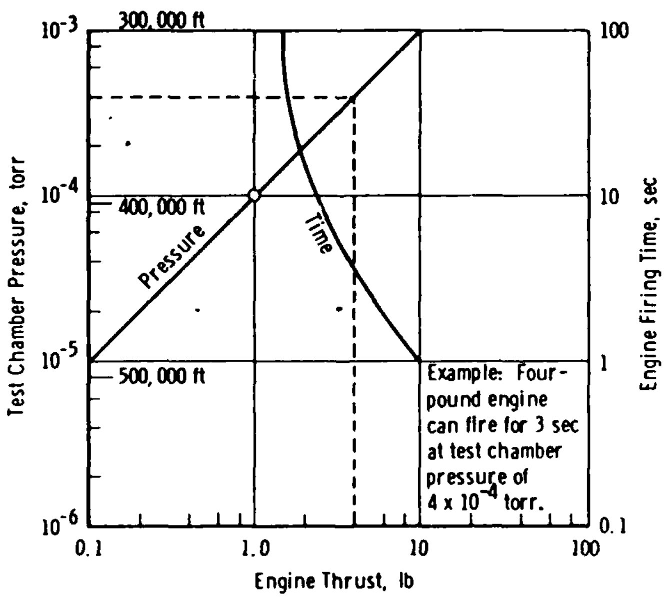


Fig. 10.5 Performance of Aerospace Chamber (10V) for Space Propulsion Testing

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